# Monte Carlo Analysis as a Trajectory Design Driver for the Transiting Exoplanet Survey Satellite (TESS) Mission

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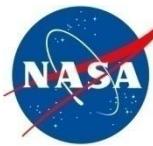
Overview

- **The TESS mission utilizes a 2:1 Lunar Resonant orbit similar to IBEX and phasing loops similar to LADEE**

- **To design the TESS trajectory we draw on understanding of 3- and 4-body dynamics gained from both missions and from some smart people**

- **For TESS a major trajectory design challenge is to mitigate risk so that statistical DV is not too large, and so that a missed maneuver in the phasing loops would not cause us to lose the mission**

- **There are also Sun angle constraints on maneuvers, perigee altitude constraints and eclipse constraints that complicate the design**

- **The team of GSFC and ADS Flight Dynamics engineers has developed scripts using the General Mission Analysis Tool (GMAT) with distributed processing in a generic software architecture to efficiently design trajectories and to run Monte Carlo simulations, with limited user intervention required**

- **With this architecture we ensure that mission constraints are met and we can explore the mission design trade space**
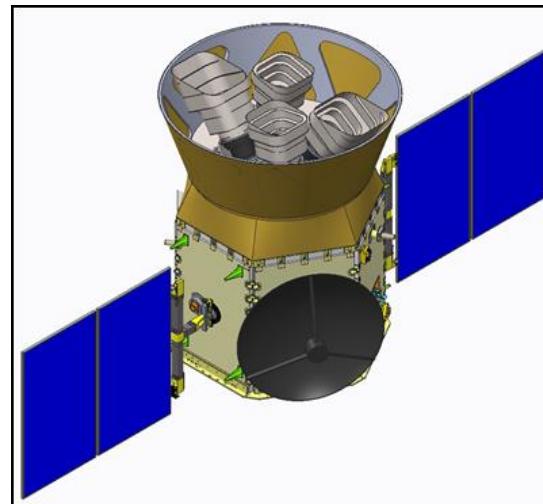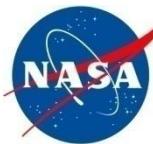
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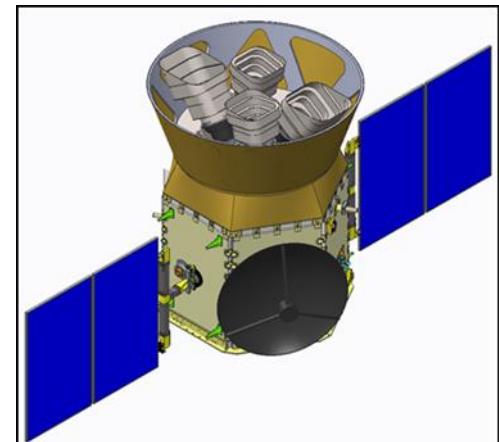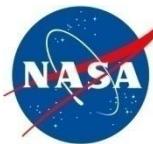
Launch on SpaceX Falcon 9
Launch in Dec 2017

Phasing loops 2 & 3
variable by launch date

Green: Phasing loops
Magenta: transfer orbit after flyby
Gold: Mission orbit

Moon

TESS at PLEP

Earth

Lunar flyby
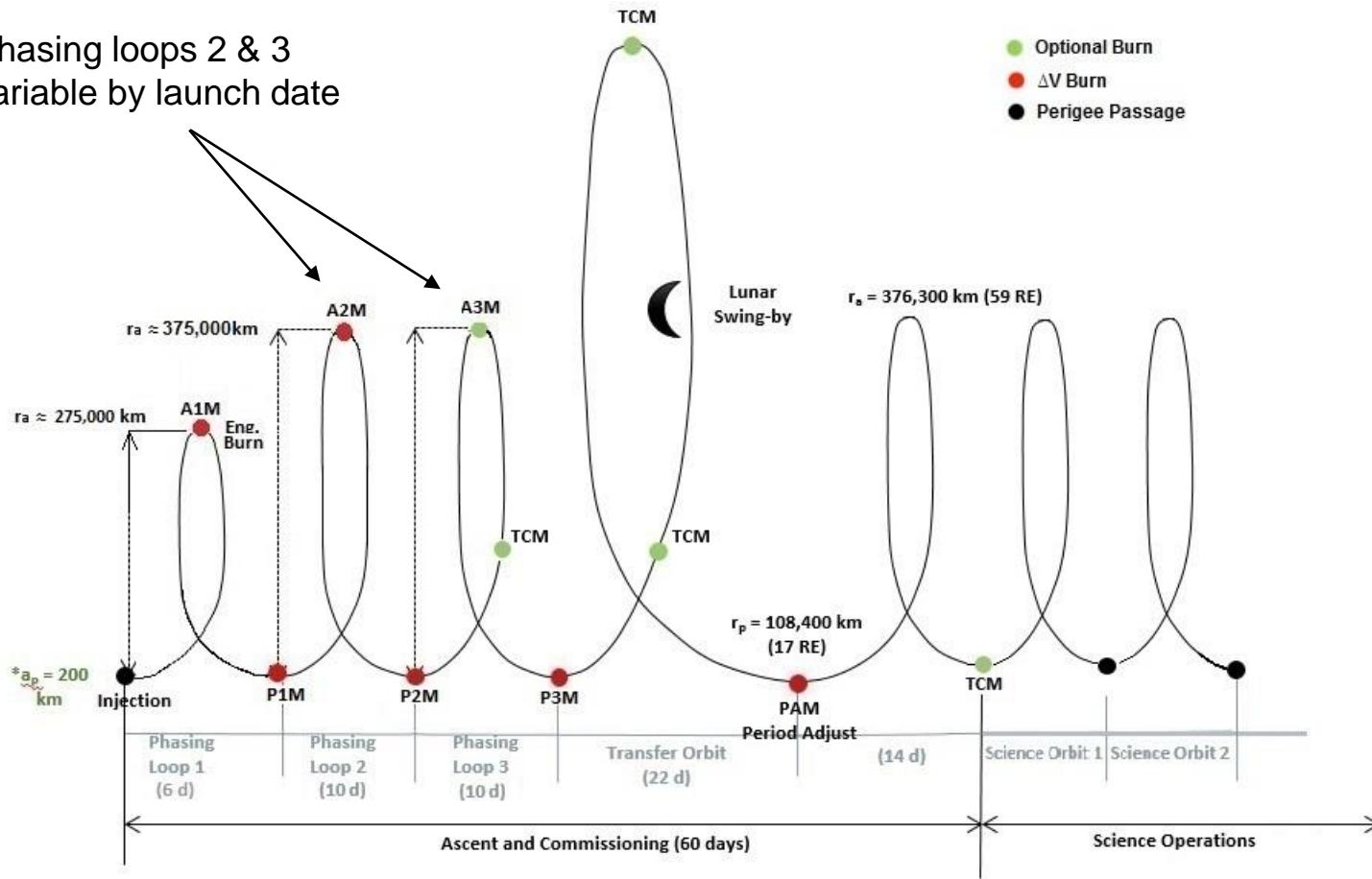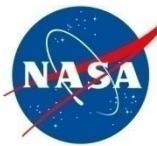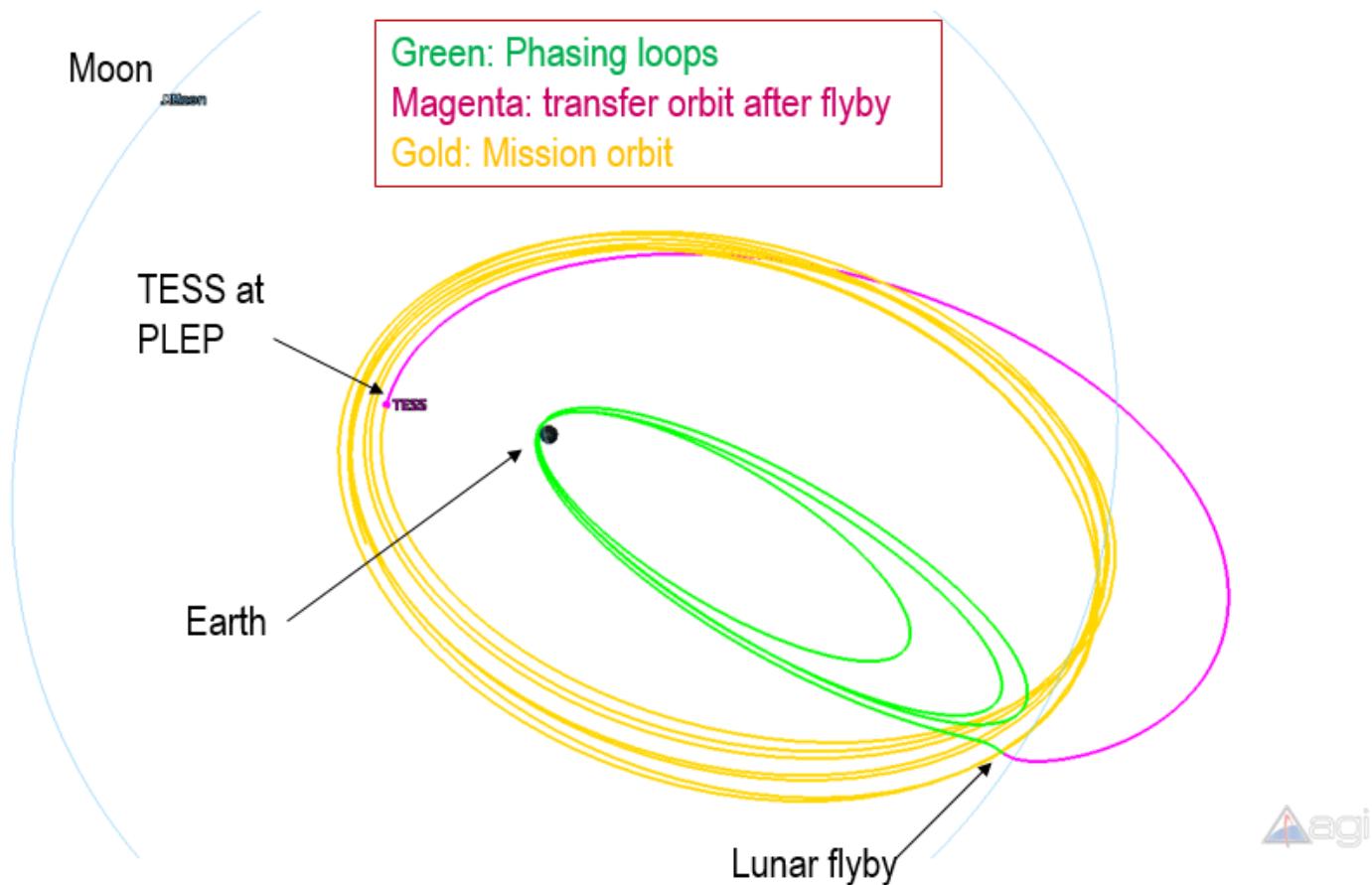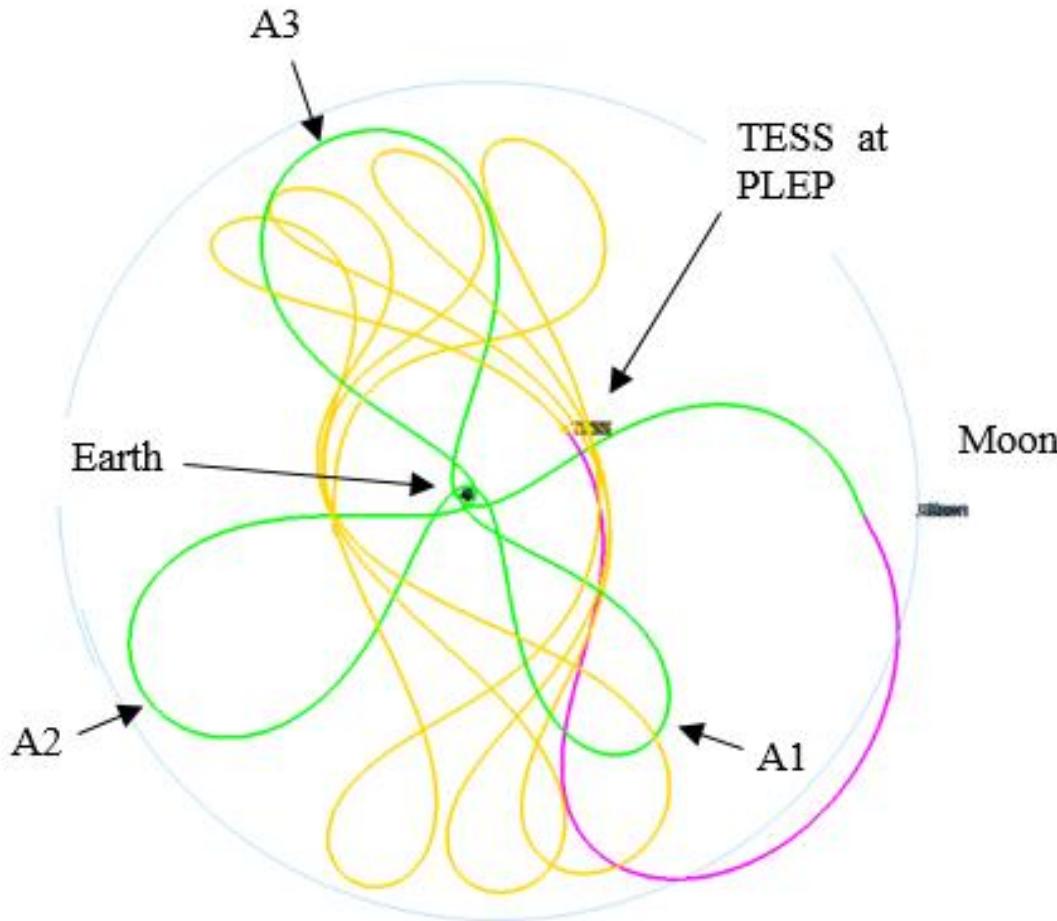
Lunar Resonant Phasing (LRP) condition: At apogee the Moon-Earth-spacecraft angle is large. This removes the need for orbit maintenance maneuvers and makes the orbit operationally stable

The phasing loop duration is chosen so at A1, the Moon is ahead of the spacecraft. This raises perigee naturally, so A1M is not critical for the mission
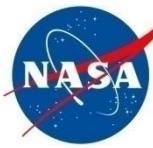
Mission constraints

- **Mission orbit period 2:1 resonant with Moon (13.65 days)**
  - **Semimajor axis 38 Re**
- **At least five launch opportunities in a lunar cycle**
- **Achieve mission orbit within 2 months after launch**
- **Total DV <= 215 m/s**
  - **~150 m/s deterministic, ~25 m/s for injection error, ~20 m/s for statistical DV, ~20 m/s margin**
- **Initial mission orbit perigee = 17 Re**
- **Mission orbit perigee remains below 22 Re for communications**
- Eclipses
  - At most 16
  - No more than two between 4 to 5 hours; the rest less than 4 hours
- Phasing loop perigee P1, P2, P3 above 600 km
- **Sun angle at maneuvers: Sun-to-boresight angle can be less than 30 deg for no more than 15 minutes at a time**
- Mission orbit perigee stays above GEO radius for 30 years

Grayed-out constraints are checked but not currently enforced in Monte Carlo simulation
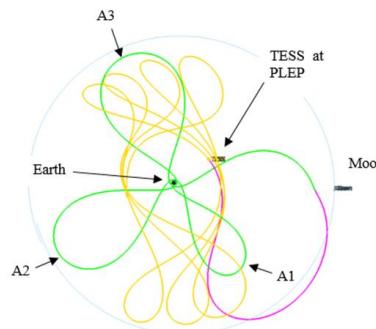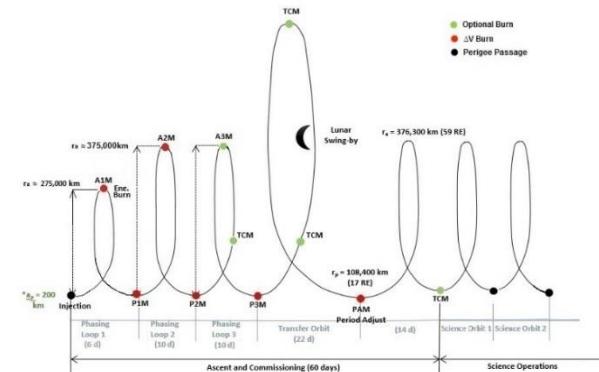
- **Use lunar flyby to achieve transfer orbit plane change, apogee raise and perigee raise**

- **Use gravity assist from the Moon near A1 to raise P1 radius with no DV cost**

- **Whenever practical, use ideal phasing loop duration so P2M and P3M are near zero.**
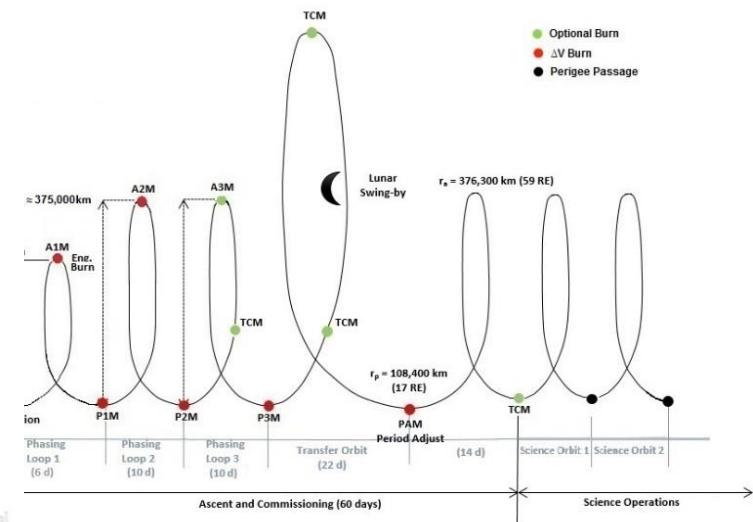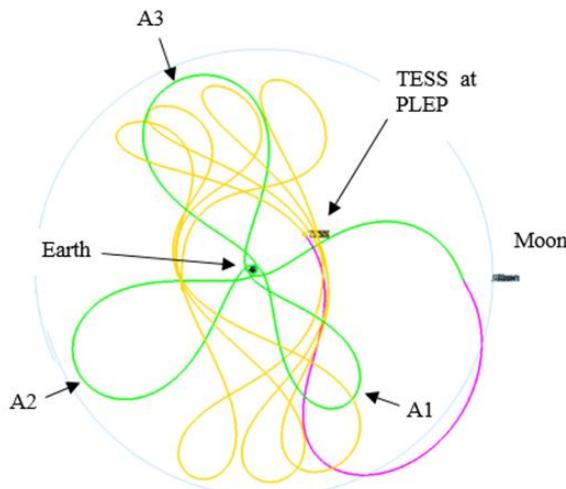
Maneuver Replanning Method: Lambert Solution

- **One fundamental approach to replan maneuvers is a Lambert solution:**
  - After a perturbed maneuver, wait sufficient time (~1 day) to get new state estimation
  - Plan a Trajectory Correction Maneuver (TCM) to return to the target position on the reference trajectory at a desired epoch
  - When the target epoch is reached, perform a 2nd maneuver to achieve the target position and velocity
- **Advantages of the Lambert solution**
  - We return to the reference trajectory that meets all mission constraints and goals.
  - After return to the reference trajectory, no further maneuver replanning is needed
- **The Lambert solution is well-suited to a case where dynamics are close to two-body motion**
- **In general we do not use a Lambert solution**
  - We found for TESS that significant perturbations from the Moon and Sun, a Lambert solution can require 80-100 m/s DV to correct for a realistic error in a single maneuver
  - The mission does not require that we return to a reference trajectory
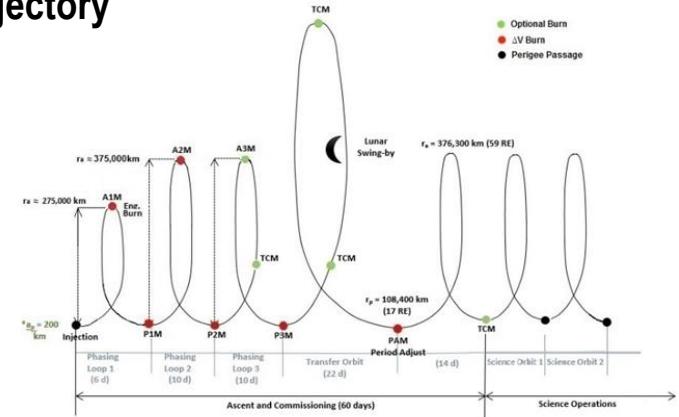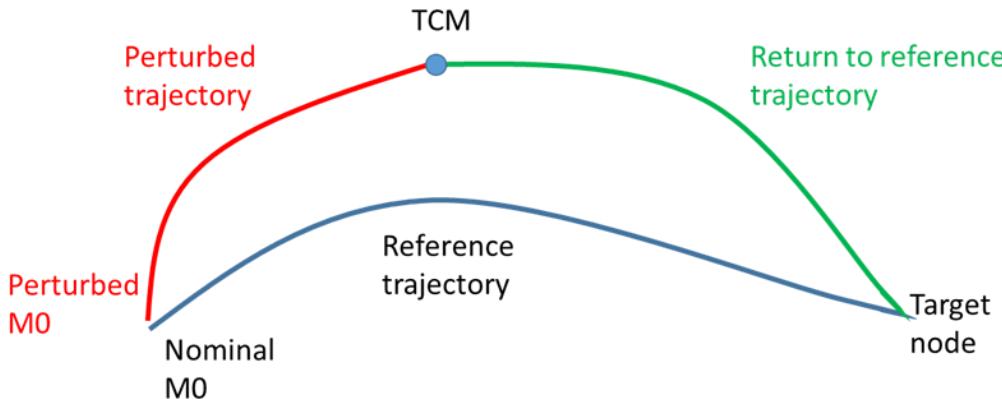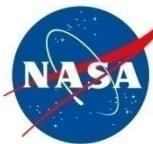
Maneuver Replanning Method (cont'd)

- **Because we do not return to a reference trajectory, after each perturbed maneuver we replan all subsequent maneuvers**
- **The key driver in maneuver replanning is than we perform the flyby in much the same way it was planned**
  - **We perform the flyby within about an hour of the nominal epoch**
  - **We achieve nearly the same flyby geometry, expressed in terms of the B-plane parameters**
- **Therefore we focus first on timing:**
  - **Reach P3 very close to the nominal epoch**
  - **When correcting for injection error, we initially plan to perform P1 and P2 at nominal epochs. However we may vary P1 and P2 epochs to optimize DV, with the constraint that we reach P3 at nominal epoch**
- **Because we may not be at nominal P3 state at P3 epoch, achieving the B-plane parameters does not assure the desired transfer orbit**

- **To replan P3 maneuver, we use 'coarse' then 'fine' targeting:**
  - **Coarse: Achieve B-plane parameters approximately**
  - **Fine: Achieve transfer orbit apogee and perigee radius**
- **We are also exploring where to best use optimization**
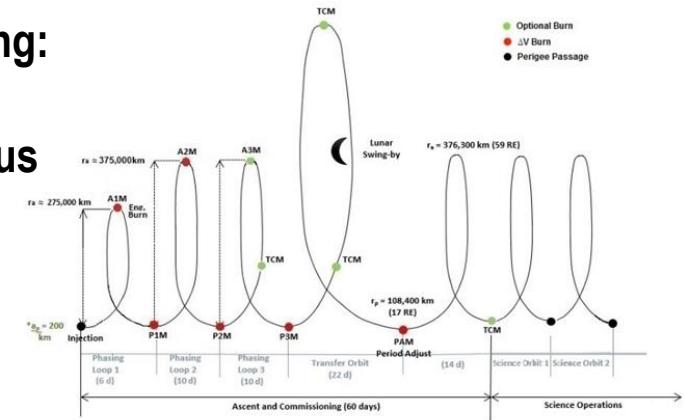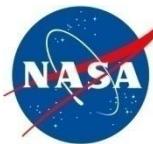
Monte Carlo Analysis Error Modeling

- **The Monte Carlo analysis is intended to provide confidence that the trajectory design can meet the mission goals and constraints, within the total Delta-V budget of 215 m/s and given the expected sources of error, in 99% of cases**
  - **400 cases was selected as a balance between statistically meaningful and runtime durations**
- **Perturbations modeled in the Monte Carlo:**
  - **Launch injection errors (primarily Apogee dispersions up to 30,000km)**
  - **Orbit Determination (OD) errors at each maneuver epoch**
  - **Maneuver execution error (magnitude and pointing)**
- **To avoid overcomplicating (and greatly increasing processing time), some constraints were not enforced or checked, instead we can include these factors in a Delta-V margin**
  - **Finite burn modeling**
  - **Launch window**

Delta-V Element	Delta-V (m/s)
Deterministic	~80-120
Launch Window	10
Finite Burn	10
Launch Injection	~30
Maneuver Execution	~10
OD Error	~5
Margin	10
Total	**< 215**

Monte Carlo DV — (Launch Injection, Maneuver Execution, OD Error, Margin)

Launch Injection Errors

- **Launch vehicle injection error is modeled based on an injection error covariance provided by SpaceX**
  - Delivered in Radius-Tangent-Normal frame, and transformed into J2000 based on nominal state
- **Injection error statistics identified in table below**
  - Allowed error in apogee radius translates to an error in the duration of the first phasing loop by up to 1 day, out of a nominal 6-day orbit

Orbit Element	Allowed Error (3σ)
Apogee Radius (km)	30,000
Perigee Radius (km)	15
Inclination (deg)	0.1
Right Ascension of the Ascending Node (deg)	0.3
Argument of Perigee (deg)	0.3
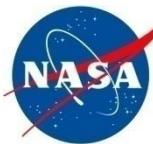
OD & Maneuver Execution Errors

- ● **Maneuver execution error is modeled in terms of both magnitude and pointing error**
 - ■ **Initial A1M magnitude error (pre-calibration) is assumed to be 5% (3σ), conservative based on previous operations experience**
  - ■ **After engine calibration, magnitude error is assumed to be 2% (3σ)**
  - ■ **Pointing error is based on data provided by the spacecraft manufacturer, Orbital ATK**
  - ■ **Values to be updated as new propulsion system data becomes available**
- ● **OD error was determined by the TESS Flight Dynamics Team using the Orbit Determination Took Kit (ODTK), developed by Analytical Graphics Inc.**
  - ■ **Realistic tracking data schedules were modeled based on agreements with the ground station providers**
  - ■ **Realistic tracking measurement statistics were used based on station-provided data and operational experience**

Maneuver	3σ Position Error (km)	3σ Velocity Error (m/s)
A1M, A2M	2	0.02
P1M	7	0.33
P2M, P3M	2	0.03
PAM	1	0.02
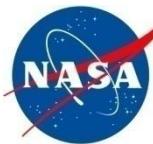
Nominal Trajectory Design Workflow

- **The primary input into the Monte Carlo analysis is the nominal trajectory design solution set**
- **The development of the nominal solution set is summarized here, greater detail is provided by the companion paper (Dichmann, 2016)**
- **The nominal trajectory design takes in all mission requirements and constraints, and develops a base set of solutions**
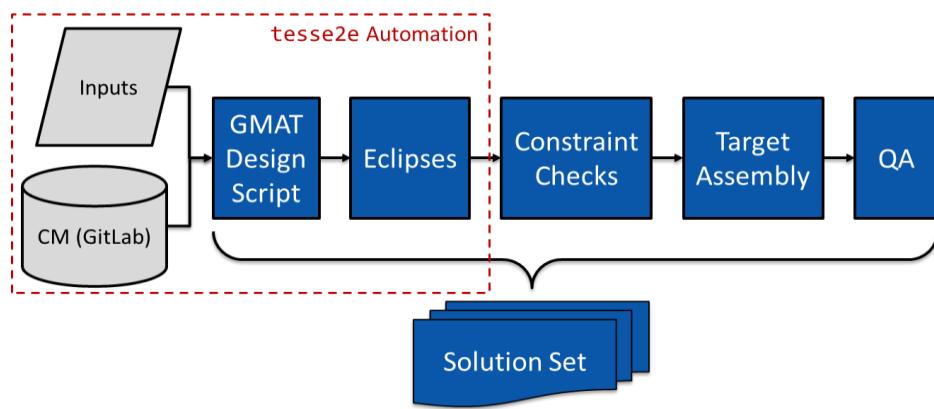
|	Dec 2017[1]	Jan 2018	Feb 2018
Attempts	19	65	57
Did not finish	(2)	(10)	(12)
Did not converge	(2)	(11)	(7)
Converged	15	44	38
Constraint: Min Perigee	(5)	(3)	(6)
Constraint: Eclipses	(3)	(11)	(10)
Constraint: Mnvr Sun Angle[2]	(0)	(0)	(0)
Constraint: Max Mnvr Size	(0)	(3)	(2)
Constraint: Deterministic DV	(0)	(0)	(0)
Constraint: TIP AOP[3]	(1)	(9)	(11)
Total Constraint Failures[4]	(8)	(18)	(22)
Feasible	7	26	16
Pruned: 1/day	(0)	(11)	(4)
Deterministic Targets	7	15	12

Monte Carlo Design

- **The Monte Carlo tool was designed to be driven and managed by Matlab**
- **Inputs:**
  - **Launch Injection Covariance (SpaceX)**
  - **OD Covariance (TESS FD Analysis)**
  - **Maneuver Execution Errors (Orbital ATK)**
  - **GMAT Trajectory Design Script (detailed on subsequent slide)**
  - **Nominal Solution Set (from previous slide)**
- **The Matlab driver:**
  - **Manages the various inputs and run settings that are available to the user**
    - **Can manage a range of potential launch dates to serially assess**
  - **Initiates parallel GMAT runs**
  - **Compiles results data from all runs for easy analysis and comparison**
 - **Optionally runs additional summary and constraint tasks**
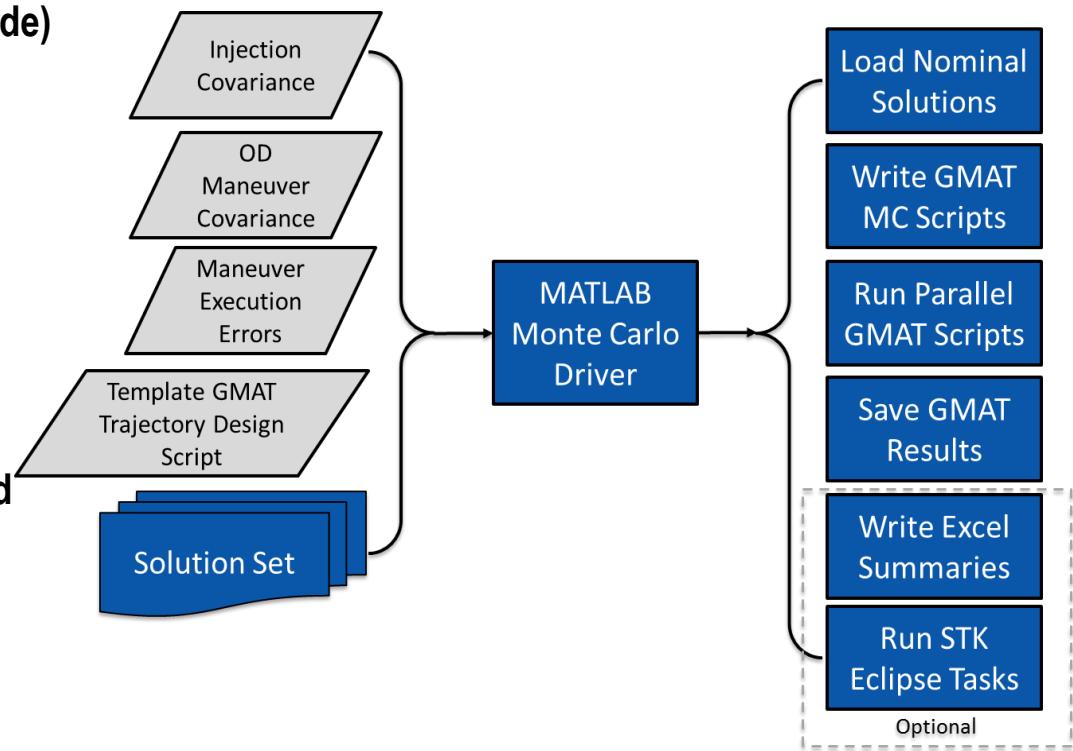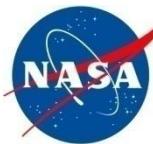
Monte Carlo Workflow

- **The GMAT Trajectory Design script utilized by the Monte Carlo analysis applies the perturbations at each stage of the trajectory, and replans the remaining trajectory to achieve a near-nominal flyby, then achieve resonant mission orbit**
- **The GMAT script serially applies the perturbations and replans the remaining trajectory to target the nominal lunar flyby**
  - **First it applies the input launch dispersion errors and replans the trajectory**
  - **Then it propagates to P1M, applies the OD and maneuver errors, and replans the trajectory**
  - **This process repeats for each maneuver, and includes TCMs for P2M, P3M and PAM**

MATLAB Monte Carlo Driver

TESS Trajectory Planning GMAT Script

- Inject Launch Dispersion Errors → Replan P1, P2, P3, TCM & PAM
- Inject OD & Mnvr Execution Errors for P1 → Replan P2, P3, TCM & PAM
- Inject OD & Mnvr Execution Errors for P2 → Replan P3, TCM & PAM
- Inject OD & Mnvr Execution Errors for P3 → Replan TCM & PAM
- Inject OD & Mnvr Execution Errors for TCM → Replan PAM

MC Results
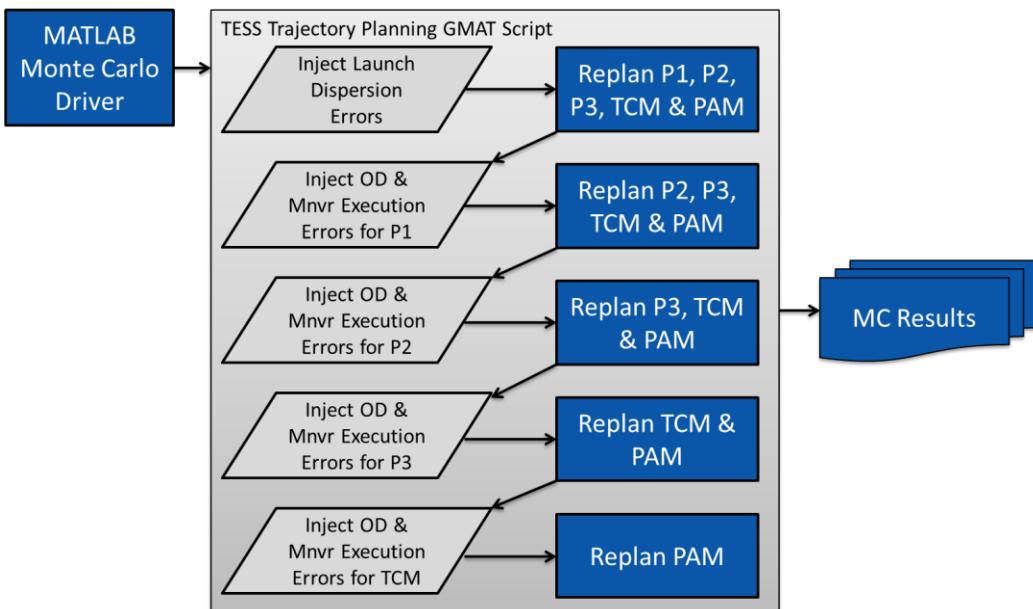
The algorithms used to replan maneuvers for MC will also be used for flight operations. The thousands of random MC trials give assurance that the flight operations will produce a trajectory that meets all mission constraints and goals
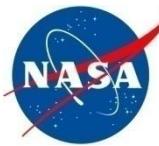
Monte Carlo Maneuver Planning

- **The trajectory design script developed for the Monte Carlo analysis will be re-used by the maneuver planning ground software (Applied Defense Solutions' Flight Dynamics System and GMAT)**

- **This greatly reduces operational risk, as the maneuver planning script has been run through thousands of potential launch, OD and maneuver perturbations**

- **The modularity of the GMAT scripts allows for additional mission operations processes to be easily integrated, such as modeling finite-burn maneuvers**

- **Validated GMAT scripts to be used for operational maneuver planning**
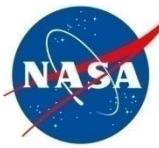
January 2018 Results

- **15 solutions were generated and validated by the nominal trajectory design process for consideration in January 2018**
- **The Delta-V breakdown for these trajectories is displayed below**
- **The Monte Carlo analysis was performed on this solution set with the following results**

January 2018 targets - ΔV
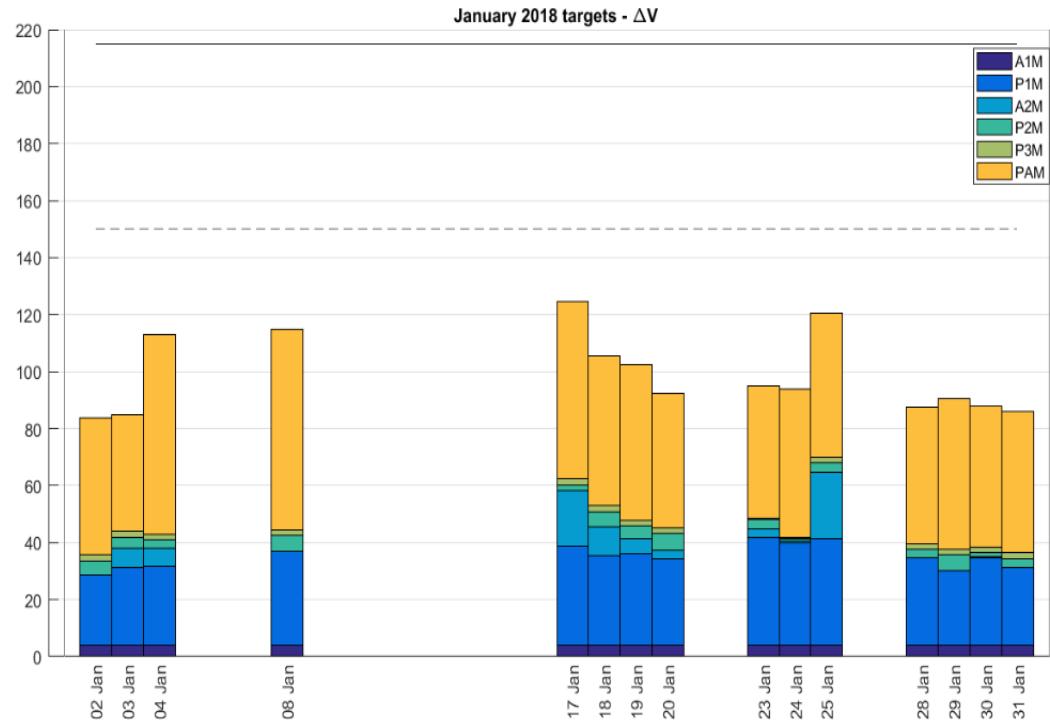
- **12 solutions passed all Monte Carlo checks and met the Delta-V requirements**
- **It is possible that the 3 nominal solutions that were lost (Jan 18-20, 2018) could be retrieved with additional date-specific maneuver planning algorithm tweaking**
- **The remaining dates meet the Delta-V constraint, with total Delta-V (including statistical) below 190 m/s**
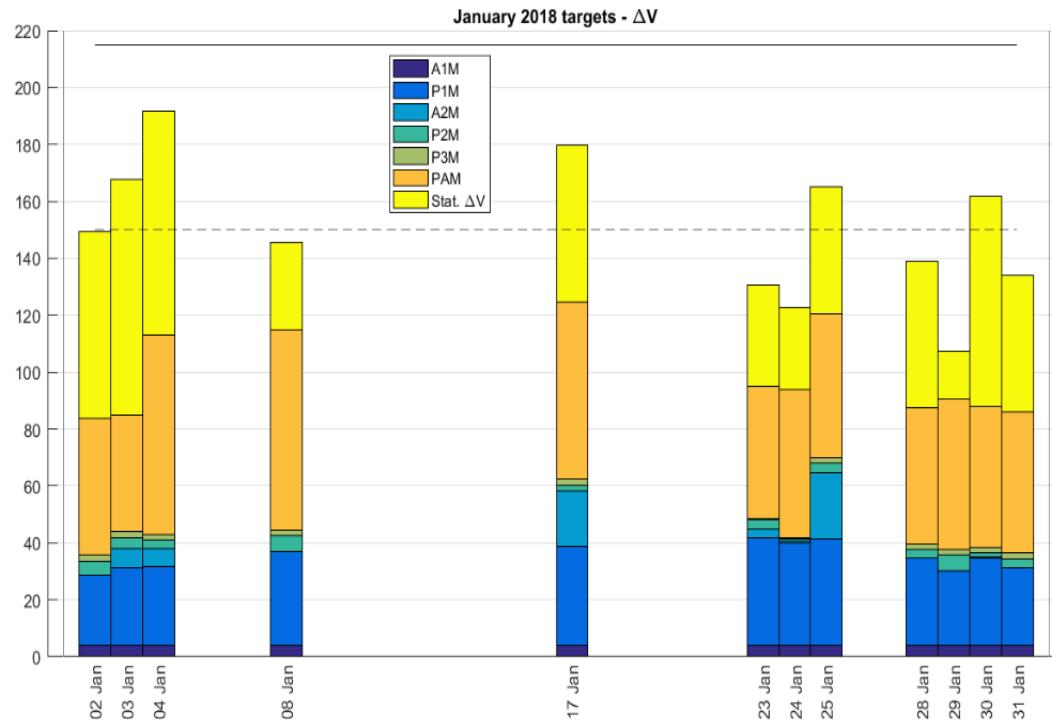
Finite Burn Assumption

- **Finite burn modeling is fairly consistent in terms of Delta-V penalty**
  - ~3% of total impulsive Delta-V (~3-7 m/s total)
  - Would add more complexity to the Monte Carlo design script without a significant benefit
 - Would non-trivially increase the runtime for each trial
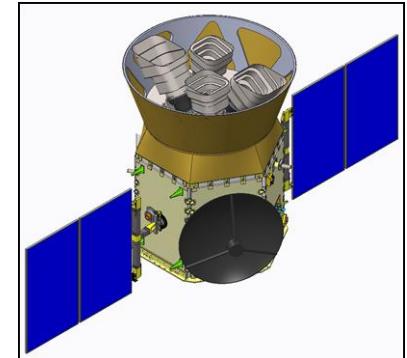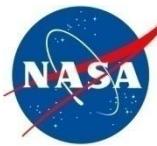
Conclusion

- **Statistical Delta-V analysis for TESS has presented various special challenges**
  - Development of maneuver planning algorithms tailored to the multibody dynamics of this trajectory
  - Algorithm robustness to all perturbation sources, primarily launch dispersions
- **Multiple planning algorithms were used to achieve greater flexibility and obtain more converged cases**
  - A great deal of analysis and stress-testing has produced a robust set of maneuver planning tools that will be utilized in flight operations
- **The large number of nominal trajectories, and subsequently required Monte Carlo trials, drove the need for a unified software architecture**
  - Architecture developed based on GMAT and Matlab
  - Manages nominal trajectory design, Monte Carlo analysis, and other specific orbit analyses
- **Simulation results for January 2018 show that we exceed the 99% Delta-V requirement for a minimum of 5 launch dates (mission requirement)**

Future Work

- Additional months validated based on PGAA-3 results, through 2018 opportunities
- With more consistent convergence, we can add in additional trajectory verification (mission duration minimum perigee, mission duration eclipse checking)
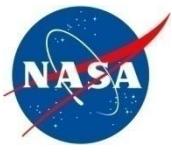
Questions
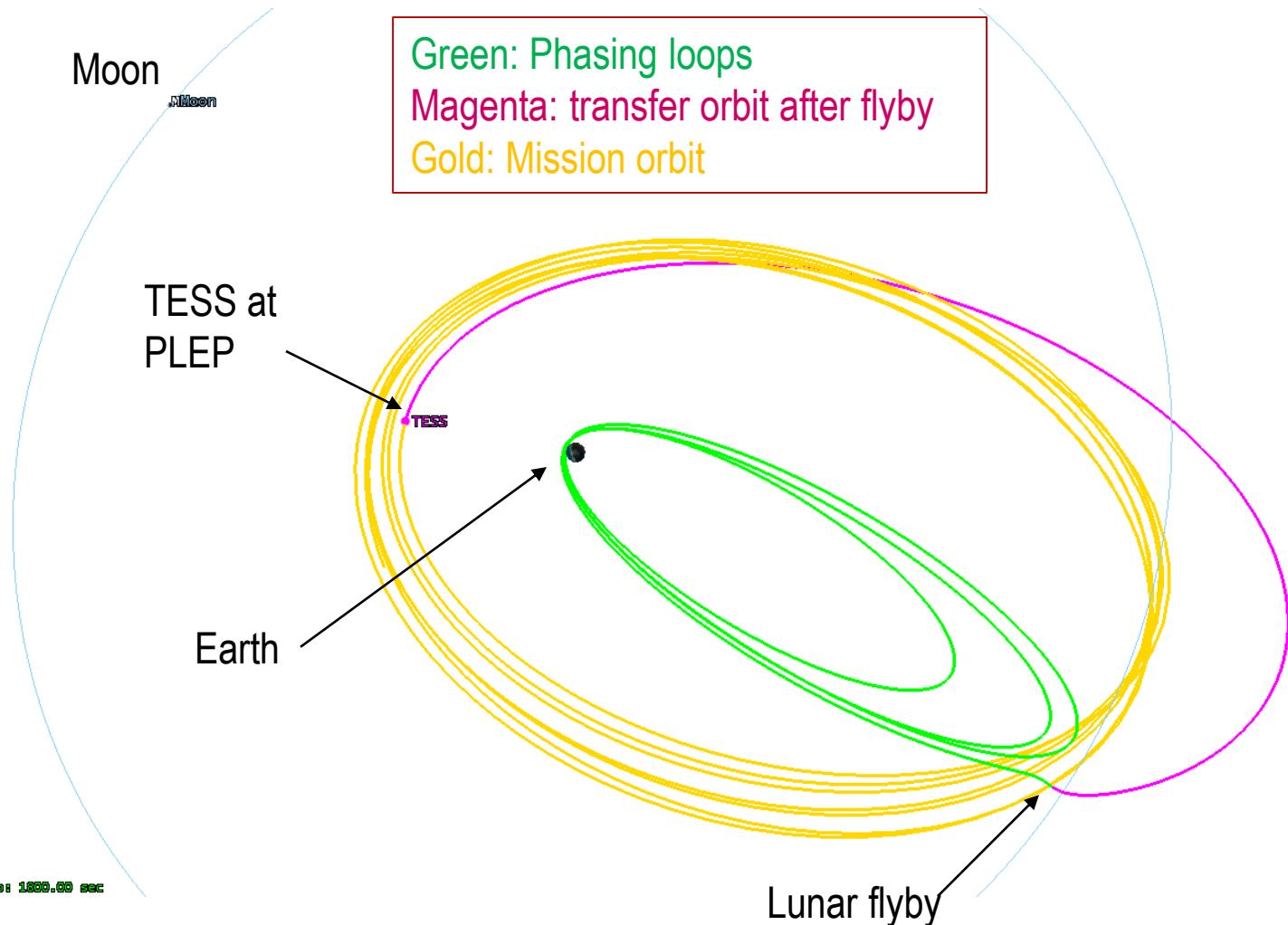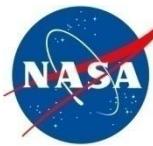
Sun Angle Constraint

- **Instrument has four cameras and large field of view, at least 30 deg from boresight**
- **Instrument (using CCDs) cannot be exposed to Sunlight for more than 15 minutes continuously**
- **PAM maneuver could be more than 15 minutes long. For P1M, inertial pointing for maneuver plus setup might last more than 15 minutes**
- **To address Sun angle constraint, we need to shift the time of a maneuver away from apsis to point along velocity vector and keep the Sun out of view**
  - **Phasing perigee maneuver may be shifted 45 minutes**
  - **Shift could cost several m/s**
  - **Phasing apogee maneuver may be shifted several hours**
  - **PAM may be shifted several hours**

Boresight and FOV for maneuver shifted from perigee

Boresight and FOV for maneuver at perigee